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Dated 28 October 2003

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APPLICATION NO.: New U.S. Application
FILED: November 21, 2003
FOR: COMPRESSOR FOR A GAS TURBINE ENGINE
ATTORNEY DOCKET NO.: 117738

03DEC02 E767895-1 D01097
P01/7700 0.00-0228050.1

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1. Your reference PAT/AT/3053

2. Patent application number *(The Patent Office will fill in this part)* 0228050.1 03 DEC 2002

3. Full name, address and postcode of the or of each applicant *(underline all surnames)*
ROLLS-ROYCE plc
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GREAT BRITAIN

Patents ADP number *(if you know it)* 3970002

If the applicant is a corporate body, give the country/state of its incorporation GREAT BRITAIN

4. Title of the invention COMPRESSOR FOR GAS TURBINE ENGINE

5. Name of your agent *(if you have one)*
MR V J BIRD

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INTELLECTUAL PROPERTY DEPT WH 58
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Patents ADP number *(if you know it)* 6578006

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| Country | Priority application number *(if you know it)* | Date of filing *(day / month / year)* |
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7. If this application is divided or otherwise derived from an earlier UK application, give the number and the filing date of the earlier application

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8. Is a statement of inventorship and of right to grant of a patent required in support of this request? *(Answer 'Yes' if:*
*a) any applicant named in part 3 is not an inventor, or*
*b) there is an inventor who is not named as an applicant, or*
*c) any named applicant is a corporate body.*
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| Description | 7 |
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| Statement of inventorship and right to grant of a patent *(Patents Form 7/77)* | NO |
| Request for preliminary examination and search *(Patents Form 9/77)* | NO |
| Request for substantive examination *(Patents Form 10/77)* | NO |
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11. I/We request the grant of a patent on the basis of this application.

Signature: V J BIRD

Date: 29 NOVEMBER 2002

12. Name and daytime telephone number of person to contact in the United Kingdom

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# COMPRESSOR FOR GAS TURBINE ENGINE

This invention relates to a compressor for a gas turbine engine

In particular a two stage compressor with a mixed flow first stage and at least one further stage for a single spool gas turbine engine.

Compressors for gas turbine engines include axial flow and radial flow types both of which have their particular advantages depending upon the specific engine application. For example, it is possible to reduce the number of compressor stages, and hence cost, by the use of a single centrifugal flow stage instead of a greater number of axial flow stages. Stage for stage much higher pressure ratios are possible with centrifugal compressors than axial flow machines and centrifugal flow compressors have greater resistance to foreign object damage by ingestion of objects in the intake air stream. Centrifugal flow compressors are more commonly found in small turboshaft type engines where resistance to such damage can be a highly relevant design consideration, for example in helicopter applications. Axial flow compressors on the other hand offer greater operational efficiency at the expense of more compressor stages, and hence components and cost, for a given pressure ratio. Axial flow compressors also have a lower frontal area than centrifugal flow compressors which can be an important consideration in aircraft applications. The increased frontal area, or envelope, associated with radial flow compressors is due in part to the dimensions of the radial flow impeller and also the requirement to position an annular diffuser radially outwards of the impeller.

The conflicting requirements of compressor operational efficiency and reduced number of compressor stages has been partly addressed by so-called mixed flow stages in which the flow through the compressor stage has both an axial and radial component so that the stage functions partly as a radial flow stage and partly as an axial flow stage. Mixed flow compressors offer a combination of the performance benefits of axial flow and radial flow compressors.

A two-stage compressor with a mixed flow first stage and a centrifugal flow second stage is disclosed in International Patent Application Number PCT/CA01/01335, in which the mixed flow rotor and centrifugal rotor are joined together on a common shaft which is supported with respect to the housing or compressor casing by means of a bearing assembly positioned at the forward (upstream end) of the two stage compressor. In this arrangement the compressor rotor stages may be considered to be cantilevered from the bearing at the forward end of the compressor and this can lead to problems in particular with rotor blade tip clearances. This arrangement also has the disadvantage that it requires a bearing support structure to be provided in the inlet region of the compressor resulting in a reduced flow cross-section, or a greater diameter inlet, or compressor frontal area, for a specific flow cross sectional area.

There is a requirement therefore for a compressor having at least a first mixed flow stage and a second mixed or centrifugal flow stage which exhibits improved tip clearance characteristics, and also a requirement for such a compressor having a reduced diameter inlet for a specific flow cross sectional area.

According to an aspect of the invention there is provided a compressor for a gas turbine engine; the compressor comprising a casing, a mixed flow stage including a first rotor and at least one further mixed or centrifugal flow stage including a second rotor axially spaced apart from the said first rotor, and bearing means positioned between the said first and second rotors for rotatably mounting the said rotors with respect to the said casing.

This arrangement has the particular advantage that the bearing and its associated support structure can be positioned away from the compressor inlet, at the rear face, that is to say the downstream side, of the first rotor so that the axial distance between the bearing and the rotor blades, or vanes, is significantly less than in arrangements where the rotors are effectively cantilevered from the front of the compressor as previously described. In this way it is readily possible to maintain an appropriate tip clearance between the compressor rotor vanes and the compressor casing or other

ducting forming the radially outer wall of the annular gas flow passage of the compressor. This arrangement has the additional benefit of reducing the compressor inlet diameter for a specific inlet flow area since the radially inner flow boundary of the inlet may be positioned closer to the rotation axis of the compressor. In this aspect of the invention the bearing and associated support structure is positioned remote from the compressor inlet and placed between the first and second stage rotors, that is to say between the front face of the first stage rotor and the rear face of the second stage rotor.

In preferred embodiments the bearing means is positioned closely adjacent to the first rotor. This arrangement has the additional benefit of readily enabling a frusto-conical bearing support structure to be positioned between the first and second stage rotors to support the bearing loads with respect to the compressor casing. This arrangement has particular advantages where the second stage comprises a centrifugal flow compressor stage where radial tip clearances are of less significance than in the first stage mixed flow impeller.

Preferably, the first and second rotors are carried on a common shaft and the bearing means rotatably mounts the shaft with respect to a bearing support means. By positioning the bearing between the first and second stage rotors the bearing support can be positioned downstream of the first compressor stage well away from the inlet section where it would reduce the flow inlet area for a particular diameter of inlet.

The bearing support means preferably extends radially with respect to the compressor shaft between the bearing and the casing so that the bearing loads can be readily transferred to the compressor casing, and preferably the bearing support is integrated, that is in the sense that it forms part of rather than being integrally formed with, with other non rotating components within the gas flow path of the compressor between the first and second stage rotors.

In one embodiment the bearing support means comprises a frusto-conical portion extending from the bearing towards the casing. This is particularly advantageous when

the bearing support transfers axial thrust loads from the compressor shaft to the casing. In particular embodiments an angle of 45° may be an appropriate angle for the conical portion.

Preferred embodiments comprise at least one diffuser section between the first and second stage rotors. In arrangements where such a diffuser section is provided the bearing support may be connected to the compressor casing through the diffuser section, that is to say through structural vane support struts extending radially between the inner and outer radial peripheries of the annular gas flow path of the compressor.

Preferably, the diffuser section has an inlet and an outlet with a radially inward flow section therebetween. In this way it is possible to redirect the flow at the exit of the first stage radially inwards so that the inlet to the second stage rotor is at a position radially inward of the first stage diffuser outlet. This readily enables the diameter of the compressor to be reduced in the region of the second stage at least, so that the frontal area of the compressor, as determined by the compressor casing, can be optimised, ie kept to a minimum.

In one particular embodiment the bearing comprises a journal bearing and in other embodiments additionally or alternatively a thrust bearing.

In one embodiment the compressor is a two staged mixed flow compressor having two mixed flow stages. In such an embodiment the compressor stages may have substantially equal pressure ratios preferably greater than 4:1 so that the two stage compressor has an overall pressure ratio of at least 16:1.

According to another aspect of the invention there is provided a gas turbine engine comprising a compressor having a mixed flow stage and at least one further mixed or centrifugal stage according to the compressor as hereinbefore defined in accordance with the above first mentioned aspect of the invention.

In one preferred embodiment the gas turbine engine is a single spool engine, for

example a turbofan or turbojet single spool engine. For the avoidance of doubt the term "single spool" used herein refers to an engine configuration in which all the turbine and compressor rotor stages are mounted on a common engine shaft. Such an engine configuration is particularly suitable for applications where a low cost engine is required, for example where production and maintenance costs are a more significant consideration than operating costs, for example where the engine has a limited operational life. The gas turbine engine may be an aero engine for a manned or unmanned aircraft as required.

An embodiment of the invention will now be more particularly described, by way of example, with reference to the accompanying drawing, which shows an axial cross-section view through part of a single spool gas turbine engine comprising a compressor according to one embodiment of the present invention.

Referring to the drawing a single spool gas turbine engine comprises a two stage mixed flow compressor 10 including a first stage 12 and a second stage 14 mounted within a generally cylindrical compressor housing 16. The compressor 10 comprises part of a single spool gas turbine engine, the other components of which are not shown in the drawing but briefly comprise a combustor downstream of the compressor and a turbine downstream of the combustor and rotatably connected to the compressor rotor stages by a common shaft 18. The shaft 18 is rotatably mounted with respect to the compressor housing or casing 16 by a first bearing assembly 20 located in the compressor, and additionally a second bearing assembly (not shown) downstream of the compressor, for example in the turbine section of the engine. The compressor 10 has an inlet 22 through which air is inducted into the first stage 12, and an outlet 24 through which high pressure air is delivered to the engine combustor section (not shown) downstream of the compressor.

The first compressor stage 12 comprises a rotor 26 which is connected to the forward end of the shaft 18 and a diffuser 28. The rotor 26 includes a disc part 30 which carries a plurality of circumferentially spaced vanes 32 which pressurise the inducted air in the annular gas flow passage defined between the rim 34 of the disc and an

interior wall part 36 of the gas flow passage. The mixed flow rotor turns the inducted inlet air so that the air which has a generally axial flow direction at the inlet has an outlet flow direction which includes a radial as well as an axial component. As it is well understood in the art of gas turbine engine design the diffuser 28 functions to reduce the outlet velocity of the gas flow exiting the rotor vanes 32 so that the flow may be delivered at an appropriate velocity to a downstream compressor stage. In the embodiment shown in the drawing the diffuser 28 turns the gas flow exiting the rotor vanes 32 so that it exits the diffuser at 38 having a substantially axial flow direction.

The second mixed flow compressor stage 14 is substantially the same as the first compressor stage in the sense that it comprises a rotor 42 including a disc 44 and vanes 46 and a diffuser section 48 immediately downstream of the rotor vanes 46. The two mixed flow stages 12 and 14 are spaced apart along the compressor axis 50 with the outlet 38 to the first stage diffuser 28 being connected to the second stage rotor inlet 52 by a further duct 40 which extends between the two compressor stages, and to a bypass duct 66 radially outwards of the duct 40. The inner and outer annular surfaces of the duct 40 comprise inflection curves of revolution such that the duct 40 turns the flow radially inwards from the diffuser outlet 38 to the second stage inlet 52 which has a smaller mean radius with respect to the compressor arcs 50 than the outlet 38. The duct 40 preferably comprises an array of support vanes 54 which extend radially through the gas flow passage of the duct to provide a structural support between the casing 16 and a bearing support assembly 56 which extends from the vane structure 54 to the bearing assembly 20. The support vanes 54 may extend only partially along the duct 40 as shown in the drawing, or alternatively extend along the full length of the duct between first stage outlet 38 and second stage inlet 52. In the embodiment shown in the drawing the vanes 54 are connected to the compressor casing through the bypass duct 66 by corresponding radially extending bypass support vanes 62.

The bearing support 56 includes a frusto-conical part 58 and a cylindrical part 60 and two radial parts 62 and 64. The annular structure defined by the bearing support parts 58, 60, 62 and 64 readily supports the shaft and rotor stages connected thereto and

provides for the transfer of engine loads from the rotatable components to the compressor casing 16. The radial parts 62 and 64 schematically illustrate the axial extent of the vanes 54 when they extend the full length of the duct 40.

In the drawing the bearing assembly 20 is positioned between the compressor rotors 26 and 42 and in this particular embodiment it is positioned towards the rear face, that is to say the downstream side, of the first stage rotor 26. In other embodiments the bearing assembly 20 may be positioned anywhere between the rotors 26 and 42 without significantly affecting the tip clearance control of the rotor vanes 32 with respect to the outer wall 36 of the annular gas flow passage of the compressor.

Although aspects of the invention have been described with reference to the embodiments shown in the accompanying drawing, it is to be understood that the invention is not limited to this precise embodiment and that various changes and modifications may be effected without further inventive skill and effort. For example, the second compressor stage 14 may comprise a centrifugal flow compressor instead of a second mixed flow compressor. In addition the gas turbine engine may comprise a turbofan or turbojet engine. In alternative embodiments the bearing support structure 56 may be supported by the compressor casing 16 through the diffuser section 28 rather than the duct 40. Moreover, one or more additional bearing assemblies may be provided between the compressor rotors 26 and 42 if additional support is required. The invention also contemplates embodiments where further mixed flow or centrifugal flow compressor stages are provided, that is to say three, four, or more compressor stages.

## CLAIMS

1 A compressor for a gas turbine engine; the compressor comprising a casing, a mixed flow stage including a first rotor and at least one further mixed or centrifugal flow stage including a second rotor axially spaced apart from the said first rotor, and bearing means positioned between the said first and second rotors for rotatably mounting the said rotors with respect to the said casing.

2 A compressor as claimed in Claim 1 wherein the said bearing means is positioned closely adjacent the said first rotor.

3 A compressor as claimed in Claim 1 or Claim 2 wherein the said first and second rotors are carried on a common shaft and the said bearing means rotatably mounts the said shaft with respect to a bearing support means.

4 A compressor as claimed in Claim 3 wherein the said bearing support means extends radially with respect to the said shaft between the said bearing means and the said casing.

5 A compressor as claimed in Claim 4 wherein the said bearing support means comprises a frusta-conical portion extending from the said bearing means towards the said casing.

6 A compressor as claimed in any of Claims 3 to 5 further comprising at least one diffuser section between the said rotors and wherein the said bearing support means is connected to the said casing through the said diffuser section.

7 A compressor as claimed in Claim 6 wherein the said diffuser section has an inlet section and an outlet section with a radially inward flow section there between.

8 A compressor as claimed in any preceding claim wherein the bearing means comprises a journal bearing.

9 A compressor as claimed in Claim 8 wherein the bearing means additionally comprises a thrust bearing.

10 A compressor as claimed in any preceding claim wherein the said compressor is a two stage mixed flow compressor having two mixed flow stages.

11 A compressor as claimed in any preceding claim wherein the said compressor stages have substantially the same pressure ratio.

12 A compressor as claimed in Claim 11 wherein the pressure ratio of each stage is greater than 4:1.

13 A gas turbine engine comprising a compressor according to any preceding claim.

14 A gas turbine engine as claimed in Claim 13 wherein the said origin is a single spool engine.

15 A gas turbine engine as claimed in Claim 14 wherein the said engine comprises a turbo-fan or a turbo-jet engine.

16 A gas turbine engines as claimed in Claims 13 to 15 wherein the said engine is for an aircraft.

17 A compressor substantially as hereinbefore described and/or with reference to the accompanying drawings.

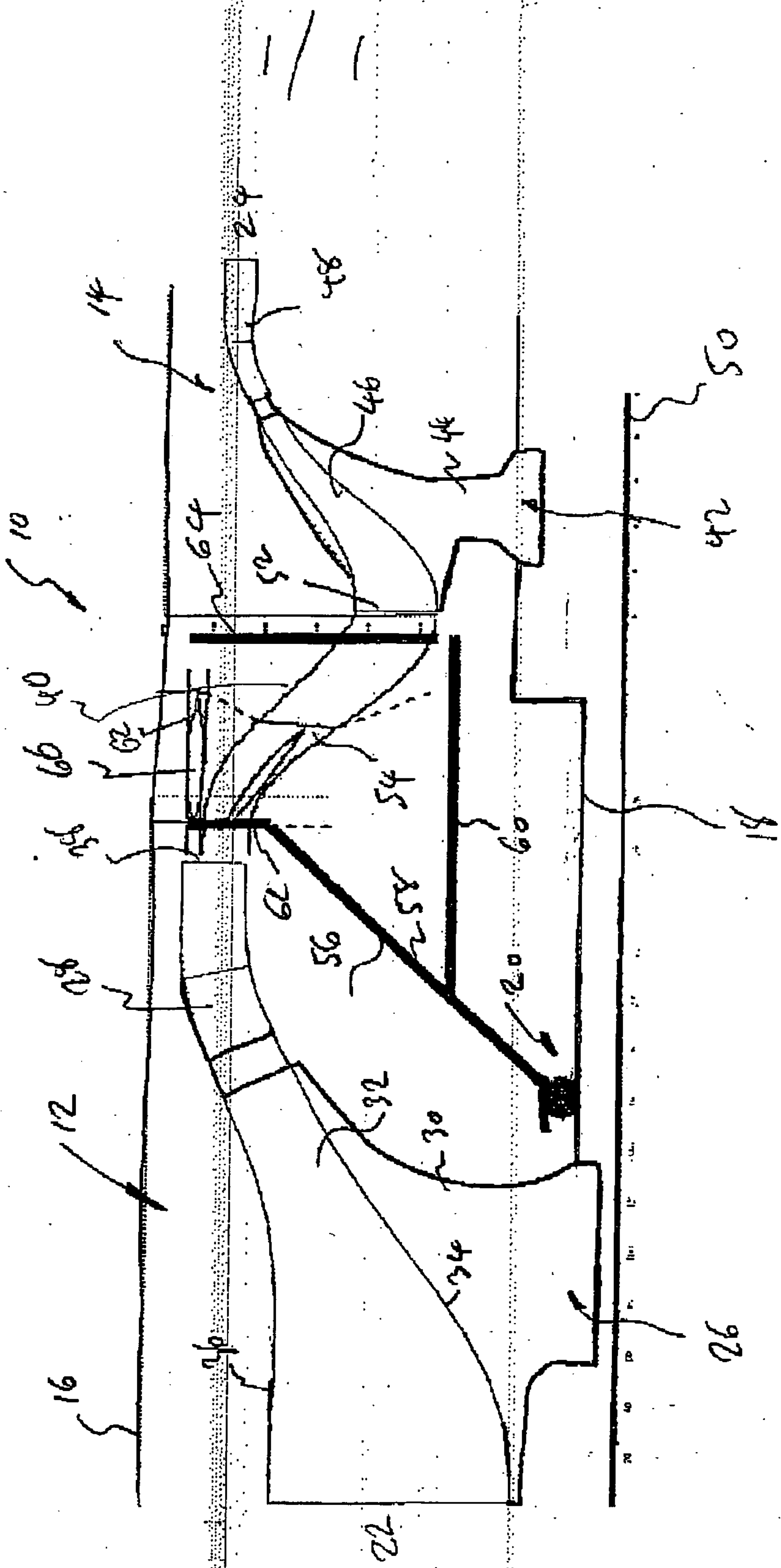
18 A gas turbine engine substantially as hereinbefore described and/or with reference to the accompanying drawings.

## ABSTRACT

## COMPRESSOR FOR GAS TURBINE ENGINE

A compressor for a gas turbine engine comprising a casing, a mixed flow stage including a first rotor and at least one further mixed or centrifugal flow stage including a second rotor axially spaced apart from the first rotor, and bearing means positioned between the first and second rotors for rotatably mounting the rotors with respect to the casing. The bearing means is positioned closely adjacent the first rotor.

The rotors are carried on a common shaft and the bearing means rotatably mounts the shaft with respect to a bearing support which extends radially with respect to the shaft between the bearing and the casing. The bearing support comprises a frusta-conical portion extending from the said bearing means towards the said casing.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